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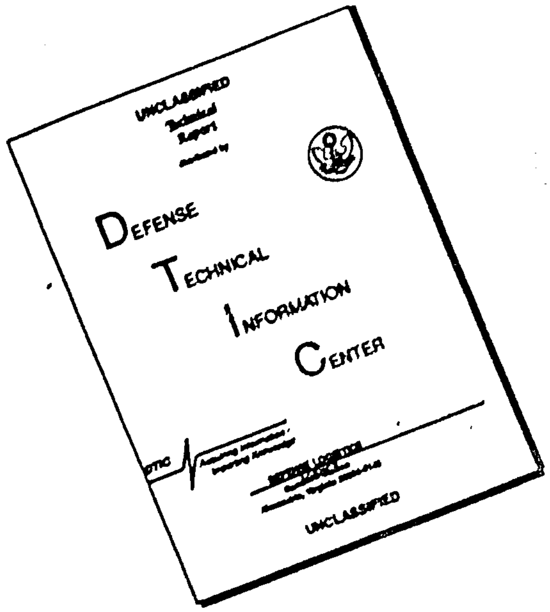

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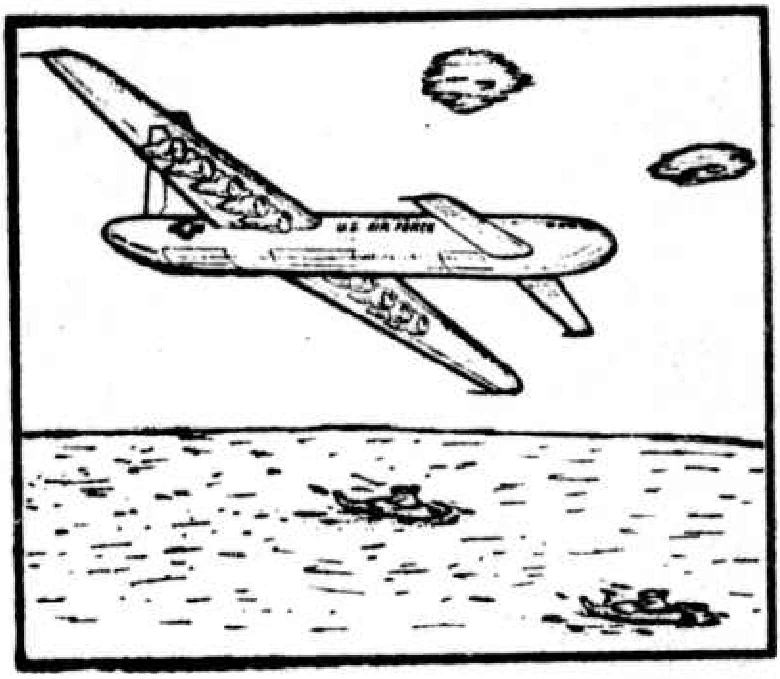
NUCLEAR AIRCRAFT FEASIBILITY STUDY

EXECUTIVE SUMMARY



1975

GRADUATE CLASS

OF

SYSTEMS ENGINEERING

MARCH 1975

SCHOOL OF ENGINEERING
AIR FORCE INSTITUTE OF TECHNOLOGY
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

NOTICE

This effort was accomplished for the purposes of illuminating problem areas in the context of a total weapon system concept and assessing the impact of different propulsion system design approaches upon the total aircraft system weight and performance. It was performed as a thesis research effort by AFIT students, and the resultant weapon system design concept is the product of the design constraints selected by the students. The influences of two of these constraints, the fuselage volume alloted per crew member, and the design wing loading are such that the resulting aircraft system size, weight, and power requirements are considerably larger than those obtained during previous in-house studies or those reported by other competent investigators examining similar mission requirements. Thus, it is important to note that the design constraints, subsystem tradeoffs, aircraft configuration selection and subsystem integration tasks were totally accomplished by the students and are, therefore, not to be construed in any way as reflecting the opinion or thinking of the Air Force or the Deputy for Development Planning.

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REPORT DOCUMENTATION PAGE

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(14)

1. REPORT NUMBER: GSE/SE/75-1 - Summ
2. GOVT ACCESSION NO.:
3. RECIPIENT'S CATALOG NUMBER:

(6)

4. TITLE (and Subtitle): NUCLEAR AIRCRAFT FEASIBILITY STUDY (EXECUTIVE SUMMARY).
5. TYPE OF REPORT & PERIOD COVERED: MASTERS THESIS
6. PERFORMING ORG. REPORT NUMBER:
7. AUTHOR(s): M.W. AMOS, J.S. BUTT, B.L. CAMPBELL, R.G. DIEHL, W.J. FANNING, JR., G.M. LACY, R. J. LUCZAK, E.M. MOORMAN, JR., J.W. SHATTUCK, V.K. WILKINSON
8. CONTRACT OR GRANT NUMBER(s): (12) 37p.
9. PERFORMING ORGANIZATION NAME AND ADDRESS: AIR FORCE INSTITUTE OF TECHNOLOGY (AFIT/EN) WRIGHT-PATTERSON AFB OH 45433
10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: P.E. 6 3737F
11. CONTROLLING OFFICE NAME AND ADDRESS: AERONAUTICAL SYSTEMS DIVISION (ASD/XR) WRIGHT-PATTERSON AFB OH 45433
12. REPORT DATE: (11) March 1975
13. NUMBER OF PAGES: 20
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office):
15. SECURITY CLASS. (of this report): UNCLASSIFIED

15a. DECLASSIFICATION DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Distribution limited to U.S. Gov't. agencies only; Test and Evaluation; 1 MAR 75. Other requests for this document must be referred to [redacted] of ASD/XR, Wright-Patterson AFB OH 45433.

(9) Rept. for 1 Mar 74 - 7 Feb 75,

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

(10) Michael W. /Amos, James S. /Butt
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William J. /Fanning, Jr

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Nuclear propulsion, fault tree analysis, event tree analysis, heat transfer, life cycle cost, nuclear aircraft, weight estimation, anti-submarine, reactor power, large aircraft design, long endurance aircraft, nuclear safety, helium cooled reactor, liquid-metal cooled reactor.

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The objective of this study was to assess the feasibility of applying nuclear propulsion to aircraft in performance of the Air Force mission. This was accomplished by using a systems approach with the system divided into six areas: (1) Mission Selection, (2) Required Mission Avionics, (3) Aircraft Design, (4) Propulsion System Design, (5) Public Safety, and (6) Cost. The overriding constraint of the study was the assumption that technology would limit an aircraft gross weight to 2,000,000 lbs in the 1990's. At this gross weight, an aircraft built using conventional construction methods and powered (SEE REVERSE)

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by a liquid metal cooled nuclear reactor, but using only chemical fuel for takeoff, would have a negative payload of 120,000 lbs. If the aircraft were constructed using advanced composites and a liquid metal cooled reactor with chemical augmentation for takeoff, the payload would be 470,000 lbs. By switching the liquid metal reactor for a similarly constructed helium cooled reactor, the payload would drop from 470,000 lbs to 210,000 lbs. For each individual in the U. S., the risk of being killed by the radioactive particles associated with one of the airborne 574 MW reactors, would be 9.34 x 10^{-8} per year, which is less risk than that of being struck by lightning. The 52,000 hr airframe life cycle cost was estimated to be $26.4 billion for 60 aircraft.

10 to the 8th power

PREFACE

This report was prepared by the 1975-M Graduate Systems Engineering Class (GSE-75M) at the Air Force Institute of Technology (AFIT), Wright-Patterson AFB, Ohio, in partial fulfillment of requirements for the Master of Science degree. This is not an official Air Force Design Study. The work was sponsored by the Propulsion and Energy Division, Deputy for Development Planning, Aeronautical Systems Division, Air Force Systems Command, Wright-Patterson AFB, Ohio. Authors of this report are the following 10 members of the AFIT GSE-75M class:

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Technical work was conducted during the period 1 March 1974 to 7 February 1975. The final report was submitted on 14 March 1975.

Special appreciation is given to Col. James H. Hall, Deputy for Development Planning, ASD, and Dr. Larry W. Noggle, Nuclear Propulsion Technology Program Manager, ASD, for their close support and cooperation in all phases of the study.

Special recognition is given to the following for their cooperation in providing technical assistance in all phases of the study:

- Mr. Gary Averill, Naval Air Development Center.
- Maj. George T. Babbitt, USAF
- Mr. & Mrs. William Bustard, AFIT
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The report consists of three volumes: Volume I, *Final Report*; Volume II, *Appendices*; and Volume III, *Classified Annex*.

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EXECUTIVE SUMMARY

This Executive Summary presents the results of a nuclear aircraft design feasibility study prepared by the 1975 Graduate Systems Engineering class at the Air Force Institute of Technology, Wright-Patterson AFB, Ohio, in partial fulfillment of the requirements of a Master of Science degree. The work was sponsored by the Propulsion and Energy Division of the Aeronautical Systems Division, Deputy for Development Planning.

1 BACKGROUND

The last major effort to develop a nuclear powered airplane was the Aircraft Nuclear Propulsion (ANP) Development Program which was terminated in 1961. The technology of that decade was such that the program was unable to meet its design objectives, and a nuclear powered airplane was deemed infeasible.

Since 1961, a combination of several factors has caused a renewed interest in the concept of a nuclear powered airplane. First, the United States is no longer self sufficient in meeting its energy demand, in that it is now dependent on foreign sources for a major portion of its oil needs. Several alternative forms of energy are being studied in an effort to relieve the dependence on foreign oil, with hydrogen, methane, and nuclear fuels being considered for aircraft propulsion.

Second, significant improvements have been made in nuclear technology since 1961. Of particular importance are developments in high temperature, gas-cooled and liquid metal-cooled reactors, and the production of nuclear fuels with higher power densities and longer lifetimes. Collectively, these developments have given support to the concept of a relatively lightweight nuclear power system that may be used for aircraft power.

Finally, changes in the performance requirements for a nucelar powered aircraft have made feasibility easier to attain. The earlier programs sought an aircraft with supersonic speed capability; however, this capability is no longer a prerequisite. The criteria now simply involve the replacement of fossile fueled aircraft in the performance of Air Force missions.

While these factors have changed so as to support the concept of a nuclear powered airplane, another factor, that of public safety, has changed in exactly the opposite fashion. The public is now acutely aware of the potential hazards of any nuclear power system; therefore, public safety must now be a major design consideration. This concern requires the expenditure of considerable design effort to insure that a nuclear powered airplane does not pose an unacceptable hazard to the public.

The objective of this study was to assess, through a comprehensive systems engineering analysis the feasibility of applying nuclear propulsion to aircraft in the performance of the Air Force mission

The specific major areas of study addressed in this report are: (1) mission, (2) avionics, (3) aircraft, (4) propulsion system, (5) safety, and (6) cost. The propulsion system is further divided into three sections: reactor, engines, and heat transfer system. This summary first presents the major details and the results obtained from each section of the report. Next, a systems integration is presented which is then followed by the conclusions and recommendations.

1.1 MISSION (Volume I,* Section 2): In the early phase of the study, it was determined that a large, subsonic, nuclear powered aircraft could conceivably perform several Air Force missions. These missions were: (1) antisubmarine warfare (ASW), (2) strategic airlift, (3) command and control, (4) airborne warning and control, and (5) airborne missile launch. The ASW mission was chosen as the point design mission because it readily lent itself to the operational nuclear safety considerations and because an aircraft designed for this mission could easily be adapted to perform other missions.

Mission requirements were estimated from an analysis of projected enemy submarine capabilities. Design parameters were developed based on the aircraft performance that was necessary to fulfill the mission requirements. A summary of the aircraft and mission parameters is presented in Table 1.1-1. Parameters were selected using the following bases:

1) The two week mission duration is based on human factors. Studies predict that fatigue, boredom, vibration, etc., would impair flight safety after two weeks (Volume I, Section 2).

2) The crew component of 36 men assumes three shifts of 12 men each, performing mission related jobs for eight hours per day. The 12 man crew is based on Navy studies for an advanced ASW aircraft and is comprised of a four man flight crew and an eight man tactical crew.

3) The 150 to 350 kts airspeeds are based on airspeeds currently used by the Navy during ASW operations. It was anticipated that there would be no significant increase in the speed of a submarine in the time frame addressed in this report. Consequently, current Navy ASW operations speeds were deemed sufficient.

4) The requirement that the aircraft be able to operate from sea level to 30,000 ft is compatible with projected ASW sensor capabilities.

5) The 130,000 to 200,000 lbs payload is based on projected weights of avionics and expendables as outlined in Volume I, Sections 2 and 3.

6) The takeoff and landing distances insure that the nuclear aircraft will be capable of using existing Department of Defense (DoD) airfields.

* Volume I refers to the *Nuclear Aircraft Feasibility Study, Volume I, Final Report.*

7) The nuclear powered aircraft was assumed to have unlimited range; therefore, the radius of action was not of concern in this design study.

8) The fleet size of 60 aircraft is based on an assumption that no more aircraft will be built than the number of Submarine-Launched Ballistic Missile submarines allowed by the Strategic Arms Limitations talks.

9) It was assumed that at least 20 years would be required to develop a nuclear powered aircraft. The initial operational capability would then occur during the decade of the 1990s.

10) The aircraft lifetime of 62,000 hrs is based on the current high airframe time experience of large commercial transports as discussed in Volume I, Section 2.

TABLE 1.1-1. AIRCRAFT/MISSION PARAMETERS

MISSION DURATION (HRS)	336 (2 WEEKS)
CREW COMPONENT	36 MEN (3 SHIFTS OF 12 MEN)
AIRSPEED (KTS)	150 AT SEA LEVEL 250-350 AT 30,000 FEET
ALTITUDE (FT)	SEA LEVEL TO 30,000
PAYLOAD (LBS)	MINIMUM 130,000 MAXIMUM 200,000
TAKEOFF DISTANCE	CLEAR A 35 FT OBJECT IN 12,000 FEET
LANDING DISTANCE	CLEAR A 50 FT OBJECT AND STOP IN 12,000 FEET
FLEET SIZE	60 AIRCRAFT
IOC	1990-2000
LIFETIME	62,000 FLYING HOURS

1.2 AVIONICS (Volume I, Section 3): The avionic equipment required to perform an ASW mission was defined by establishing functional subsystems to meet mission requirements. The mission duration was assumed to be up to 14 days. The problem was to determine the impact on total avionic system weight when redundancy of avionic subsystems was used to achieve a 14 day mission. Weight, volume, power, and reliability estimates were obtained for each avionic subsystem. Assuming an order of magnitude improvement in mean-time-between-failure over present day systems, and by applying redundancy, the overall system

weight, volume, and power requirements were estimated as a function of time. These estimates were obtained by minimizing system weight subject to reliability constraints. The method of generalized La Grange Multipliers was used to accomplish this task. Figure 1.2-1 plots the avionic system weight as a function of mission duration, system reliability, and component reliability. The dashed lines in the figure indicate the average number of subsys-

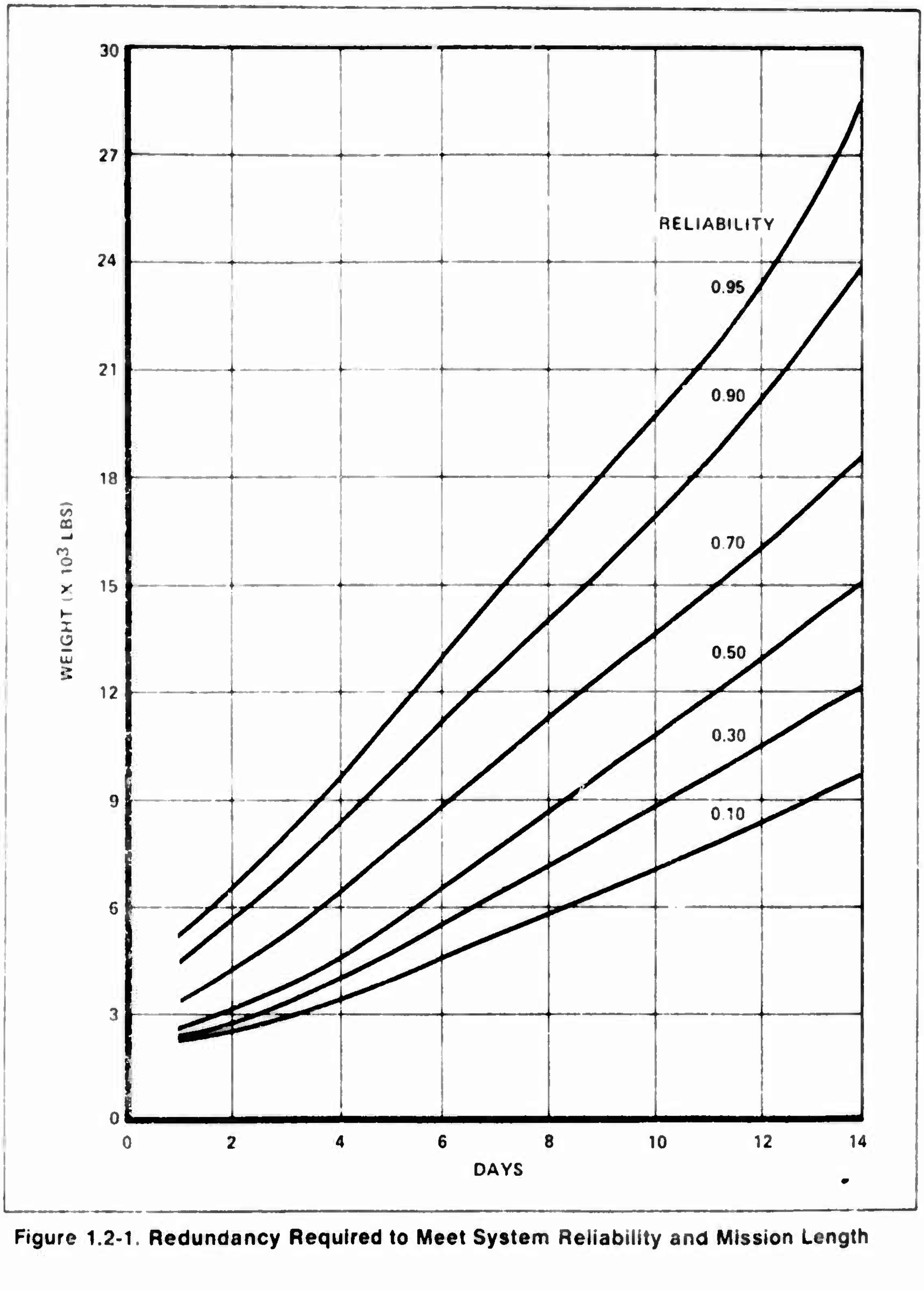


Figure 1.2-1. Redundancy Required to Meet System Reliability and Mission Length

tems which must be connected in parallel in order to meet a specified mission length and reliability. For example, if the total system were specified to be 0.90 reliable over a four day period, then the minimum system weight would be approximately 8,000 lbs. The resulting system has an average quadruple redundancy for each subsystem. Other combinations of redundancy, system reliability, and mission duration can be found in a similar manner. For a 0.95 reliable mission lasting 14 days, the weight volume, power, and average subsystem redundancy are 28,000 lbs, 600 cu ft, 240 KVA, and 10, respectively. Note that this latter volume is less than 1% of the total aircraft fuselage volume.

1.3 AIRCRAFT (Volume I, Section 4): Standard aerodynamic design analysis was used to determine an aircraft design which would fulfill an ASW mission and use nuclear power to the best advantage. Three factors, taken together, resulted in the selection of an aircraft of two million pounds gross weight for the point design. First, was an assumption that the aircraft's initial operational capability would occur during the decade of the 1990s (Volume I, Section 2). Based on empirical technology extrapolations, this time period placed an upper technology limit of 2,000,000 lbs gross weight for the aircraft. Second, was an assumption that the aircraft would operate from existing Air Force runways. This restriction also imposed a 2,000,000 lbs gross weight upper limit with respect to takeoff and landing distance and conventional landing gear capability. Finally, because of the large weight of the nuclear propulsion system (800,000 to 1,000,000 lbs), it was deemed necessary to use the 2,000,000 lbs as the design point gross weight to obtain any payload at all.

After selection of the gross weight, several configurations were considered for the point design aircraft. A canard design was selected as the configuration offering the most advantage. With the selection of the gross weight and configuration, aircraft weight and balance, dimensions and configuration, and performance were estimated.

Both aluminum and advanced composite construction were examined along with the required weight of chemical fuel to determine the weight available for the nuclear propulsion system and the payload. Table 1.3-1 summarizes the point design weight and balance for the following four options:

1) Standard aluminum construction and a chemical fuel capability for takeoff, landing, and a 500 to 1000 nm emergency cruise range.

2) Standard aluminum construction, but using a chemical fuel capability to supplement nuclear power only during takeoff and initial climb.

3) Advanced composite construction using the composite offering the best weight-saving potential. Chemical fuel is again used for takeoff, landing, and emergency cruise in this option.

4) The same composite construction as in option 3, but with chemical fuel used only to supplement nuclear power only for takeoff and initial climb.

TABLE 1.3-1. AIRCRAFT WEIGHT AND BALANCE SUMMARY — 2,000,000 LB GW

	COMPONENT WEIGHT (LBS)			
	A	B	C	D
WING WEIGHT	267,200	267,200	112,224	112,224
CANARD WEIGHT	31,200	31,200	14,664	14,664
VERTICAL STABILIZER WEIGHT	38,000	38,000	17,860	17,860
FUSELAGE WEIGHT	219,500	219,500	100,970	100,970
LANDING GEAR WEIGHT	111,000	111,000	111,000	111,000
TOTAL STRUCTURAL WEIGHT	666,900	666,900	356,718	356,718
FIXED EQUIPMENT WEIGHT	200,000	200,000	200,000	200,000
CHEMICAL ENGINE WEIGHT	50,000	14,000	50,000	14,000
CHEMICAL FUEL WEIGHT	320,000	80,000	320,000	80,000
TOTAL LESS NUCLEAR PROPULSION SYSTEM & PAYLOAD	1,236,900	960,900	926,718	650,718
AVAILABLE FOR NUCLEAR PROPULSION SYSTEM & PAYLOAD	763,100	1,039,100	1,073,282	1,349,282
CG LOCATION (FT)	265	250	260	255

A — STANDARD ALUMINUM CONSTRUCTION (VOL. I, SEC. 4.3.2) — TAKEOFF WITH CHEMICAL POWER (4.1.3)

B — STANDARD ALUMINUM CONSTRUCTION — TAKEOFF WITH NUCLEAR POWER (VOL. I, SEC. 4.1.3)

C — GRAPHITE/EPOXY ADVANCED CONSTRUCTION (VOL. I, SEC. 4.3.3) — TAKEOFF WITH CHEMICAL POWER

D — GRAPHITE/EPOXY ADVANCED CONSTRUCTION — TAKEOFF WITH NUCLEAR POWER

Table 1.3-2 details the dimensions and configurations of the point design aircraft. The 60 lbs/sq ft wing/canard loading was found to be optimum considering the mission requirements, lift-to-drag ratio, and required thrust. This wing loading yields a very large wing and a canard with almost the same area as the C-5A wing. Figure 1.3-1 shows a comparison of the point design and a C-5A aircraft.

TABLE 1.3-2. AIRCRAFT DIMENSIONS AND CONFIGURATION
2,000,000 LBS GROSS WEIGHT

WING/CANARD LOADING (PSF)	80
ASPECT RATIO	9
TAPER RATIO	0.4
THICKNESS RATIO	0.18
WING QUARTER CHORD LOCATION (FT)	300
WING AREA (SQ FT)	27,800
WING SPAN (FT)	500
WING ROOT (FT)	79
WING TIP (FT)	32
WING MAC (FT)	59
FLAP AREA (% OF WING AREA)	20
SPEED BRAKE AREA (% OF WING AREA)	13
CANARD QUARTER CHORD LOCATION (FT)	60
CANARD AREA (SQ FT)	5600
CANARD SPAN (FT)	225
CANARD ROOT (FT)	36
CANARD TIP (FT)	14
CANARD MAC (FT)	27
VERTICAL TAIL QUARTER CHORD LOCATION (FT)	360
VERTICAL TAIL AREA (SQ FT)	4300
VERTICAL TAIL HEIGHT (FT)	86
VERTICAL TAIL ROOT (FT)	70
VERTICAL TAIL TIP (FT)	30
VERTICAL TAIL MAC (FT)	53
MAX FUSELAGE DIAMETER (FT)	40
FUSELAGE LENGTH (FT)	430
REACTOR LOCATION (FT)	300
REACTOR POWER (MW)	400 — 500
LANDING GEAR LOCATION (FT)	60, 265, & 335
LANDING GEAR LOAD FACTOR (g)	3
NUMBER OF TIRES	36
TIRE SIZE	56 x 16
AIRCRAFT FOOTPRINT (SQ IN.)	8700
AIRCRAFT PRESSURE PRINT (PSI)	230

NOTE: All locations from nose of aircraft.

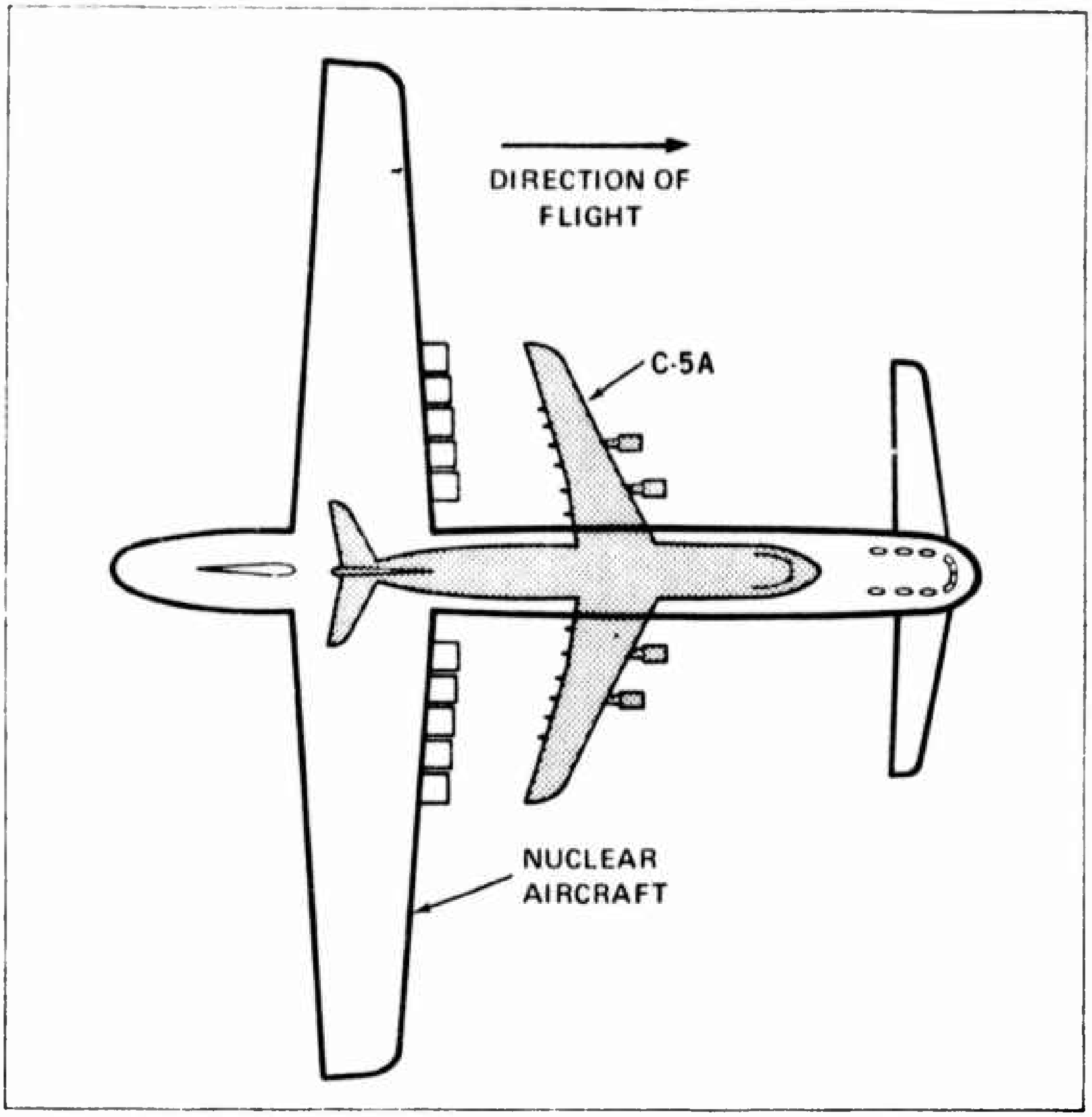
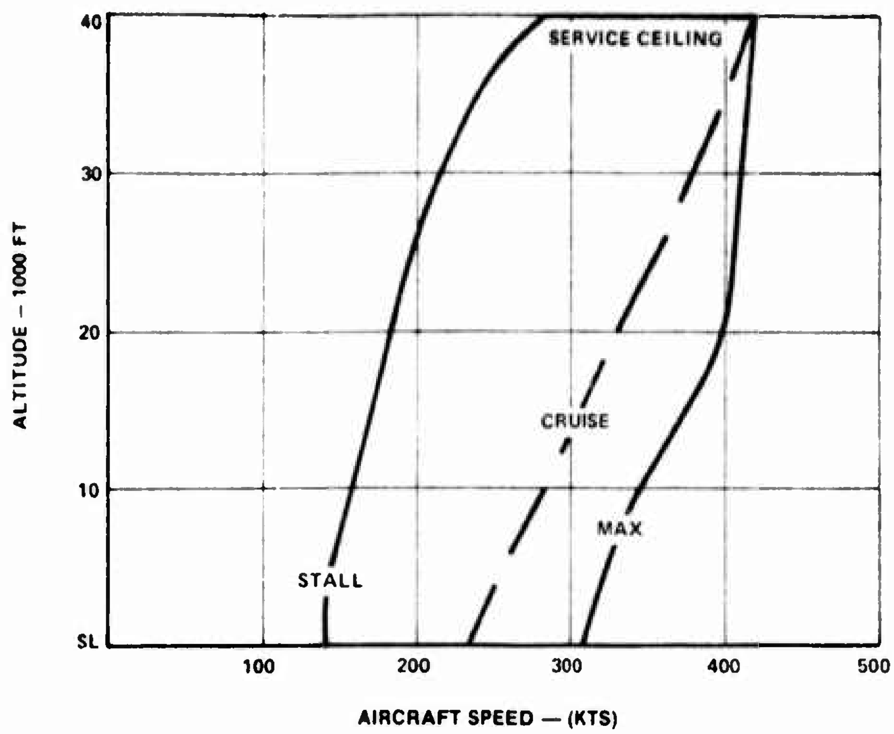


Figure 1.3-1. Nuclear Powered Aircraft - C-5A Comparison

A performance analysis summarized in Figure 1.3-2 yielded the aircraft flight envelope and the takeoff and landing parameters. The flight envelope is for a clean aircraft flying straight and level. Maximum speed is a result of total thrust available and the airfoil selected. Landing distance is nearly the same as takeoff distance, which is to be expected, since every landing is made at almost maximum gross weight. The landing deceleration calculations assumed normal wheel-braking only.



TAKEOFF THRUST	225,000 LBS
TAKEOFF DISTANCE	10,000 FT
TAKEOFF SPEED	110 KTS
MAX CLIMB RATE	600 FPM
CRUISE L/D	16.24
MAX L/D	20.30
CRUISE THRUST	123,000 LBS
MAX MACH	0.7
MAX DESCENT RATE	8000 FPM
DESCENT RATE @ TOUCHDOWN	262 FPM
TOUCHDOWN SPEED	113 KTS
LANDING STALL SPEED	98 KTS
LANDING DISTANCE	11,000 FT

Figure 1.3-2. Nuclear Powered Aircraft Performance - 2,000,000 lbs Gross Weight

1.4 REACTOR (Volume I, Section 5): The reactor study assessed the applicability of three different reactor designs to aircraft propulsion by evaluating them in three different areas. First, it was determined whether or not the required reactor power could be achieved and how much the reactor would weigh. Second, the radiation effect on the crew and the public was assessed. Third, off-point design features were analyzed to determine whether or not weight reductions were possible for the configuration used in this study.

The three reactor designs used in the study included two high temperature gas cooled reactors and one liquid metal cooled reactor. The first gas cooled design considered was the "Low Specific Weight Reactor," a 10,000 hr lifetime reactor based on the nuclear rocket

(NERVA) technology and designed by the Westinghouse Astronuclear Laboratory. The second gas cooled reactor was a 3000 hr lifetime reactor designed specifically for airborne application by King L. Mills at the University of Virginia (doctoral dissertation). The liquid metal cooled reactor was the NUERA II design by Westinghouse Astronuclear Laboratory, a 10,000 hr lifetime design using sodium-potassium (NaK) as a coolant.

Three different types of propulsion systems were proposed (see Sections I.5 and I.6), each requiring a different reactor power level. The three proposed systems are an indirect cycle liquid metal system, an indirect cycle gas system, and a direct cycle gas system, with power requirements as follows:

Liquid Metal System	475 MW
Indirect Cycle Gas System	574 MW
Direct Cycle Gas System	700 MW

Parametric data supplied by Westinghouse Corporation indicated that both their gas and liquid metal designs would supply the required power. The gas cooled design by Mills was a 200 MW point design; however, scaling techniques were applied to scale the reactor to the required power levels.

Shown in Table 1.4-1 are the reactor weights for the combinations of reactors and propulsion systems considered in the study. It can be seen that the Mills design is substantially lighter than either Westinghouse design. This is due to two factors: (1) a 3000 hr lifetime vs 10,000 hrs for Westinghouse, and (2) substantially less radiation shielding in the radial direction (perpendicular to direction of crew).

TABLE 1.4-1. REACTOR WEIGHTS (LBS)

	TYPE PROPULSION SYSTEM AND POWER REQUIRED		
REACTOR	LIQUID METAL 475 MW	INDIRECT CYCLE GAS 574 MW	DIRECT CYCLE GAS 700 MW
WESTINGHOUSE LIQUID METAL	520,000	—	—
WESTINGHOUSE HIGH TEMP GAS**	—	740,000	810,000
K L MILLS* PROPOSED DESIGN	—	459,000	504,000

* WEIGHTS INCLUDE INTERMEDIATE HEAT EXCHANGER AND 2 IN. CONTAINMENT VESSEL

** 75% EFFECTIVENESS HEAT EXCHANGER

The radiation effect of the reactors was considered for two cases: the crew and the general public. The crews were assumed to be radiation workers and therefore restricted by law to a dose rate of not more than 5 rem per year, nor more than 3 rem in any quarter. It was also assumed that the reactors would have to meet AEC standards for light water reactors, with a maximum exposure of 5 rem/year to the general populace.

As presented in Volume I, Section 4, the crew in the point design canard-type aircraft will be stationed at least 200 ft from the reactor. Taking into account only spherical attenuation, the crew will receive a dose rate of 0.05 mrem per hour for the Westinghouse reactor and 0.02 mrem per hour for the Mills reactor. Figure 1.4-1 shows the number of 14 day flights that the crew is allowed as a function of crew distance from the reactor. Note that with the canard design, the crew could be airborne all year without reaching their maximum allowable dose.

The three reactor designs can operate with the legal limits for the general populace (5 mrem/yr) if no one except the crew is allowed to remain within 4000 ft of the aircraft. The large increase in separation distance required over that of the crew is due to the radiation level in the lateral direction being as much as 6000 times that in the direction of the crew and the dose for the general populace being 1/1000 that of the crew. Figure 1.4-2 depicts the 3.5 mrem/hr isodose plot for the Mills reactor. The distance in the direction of the crew is 16 ft while the distance in the lateral direction is 800 ft.

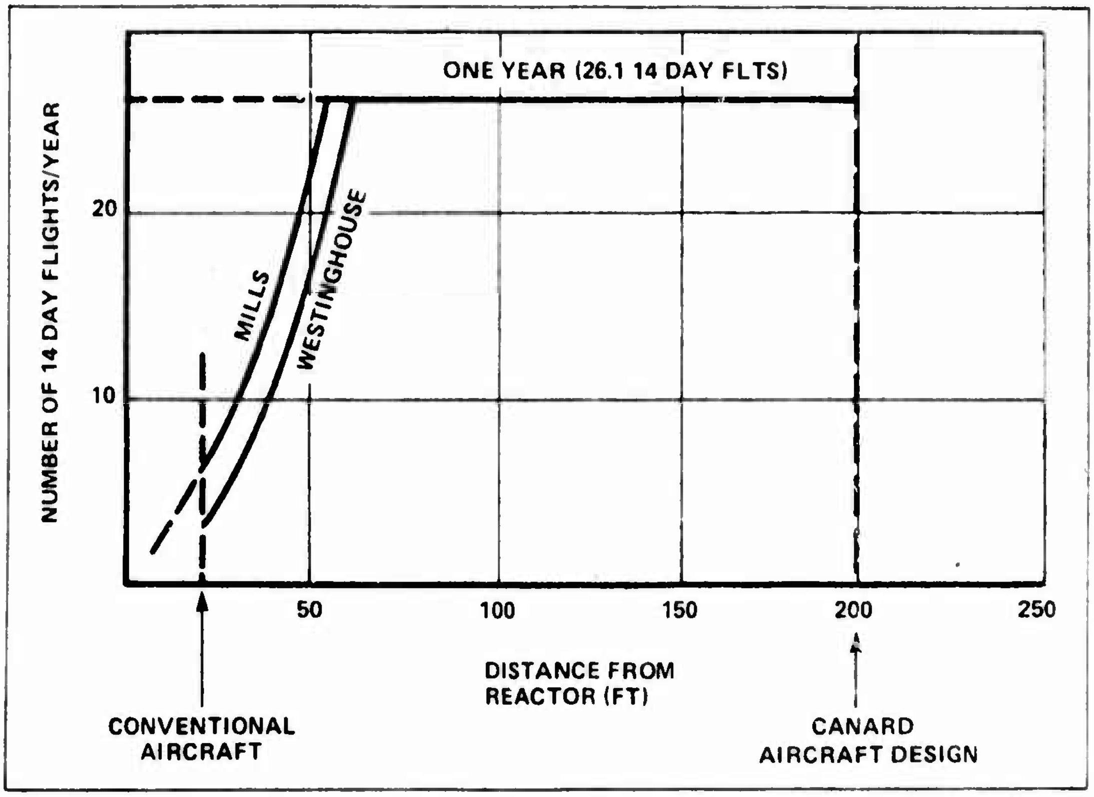


Figure 1.4-1. Number of Flights vs Distance from Reactor

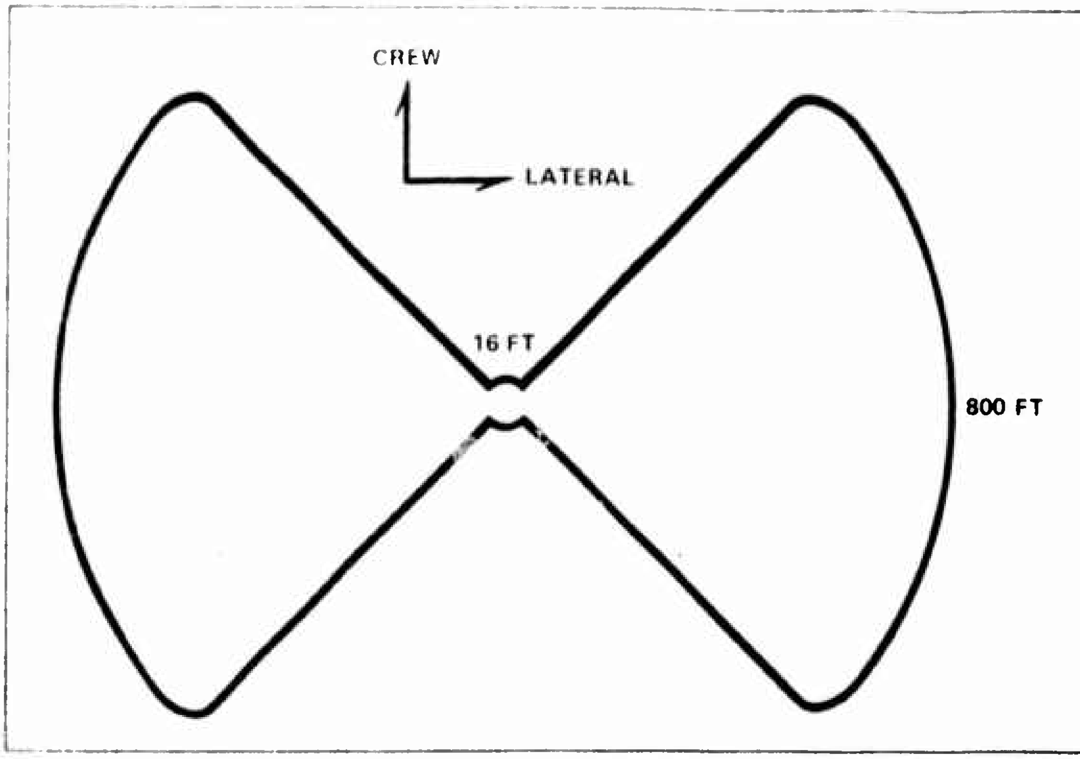


Figure 1.4-2. 3.5 Mrem/Hr Isodose

An analysis was performed to determine the amount of weight reduction possible by decreasing the shielding. The crew was assumed to fly ten 14 day flights per year of 1.49 mrem/hr. The shielding of the three reactors was then reduced until the crew would receive this much radiation at 200 ft. The results of the reduction are shown in Table 1.4-2.

TABLE 1.4-2. REACTOR WEIGHTS: ORIGINAL VS REDUCED SHIELDING

REACTOR	POWER (MW)	ORIGINAL WEIGHT (LBS)	REDUCED WEIGHT (LBS)
WESTINGHOUSE LIQUID METAL	475	520,000	N/A
WESTINGHOUSE GAS	574	740,000	719,000
	700	810,000	789,000
K L MILLS' PROPOSED DESIGN	574	452,000	442,000
	700	504,000	494,000

* WEIGHTS INCLUDE INTERMEDIATE HEAT EXCHANGER & 2 IN. CONTAINMENT VESSEL

The reactor study provided the following information. The nuclear reactors proposed are capable of supplying the power necessary. The weights of the systems can be reduced by redesigning for application to the canard type aircraft.

1.5 ENGINES (Volume I, Section 6): Two turbofan engine concepts using nuclear energy were evaluated. First, an indirect system was considered in which air as the working fluid is heated in a heat exchanger. The heat exchanger is located between the compressor exit and the turbine inlet. The second concept was a direct engine system in which helium is used as the working fluid in a closed cycle gas turbine engine, and the net useful work is used to drive a ducted fan.

In the indirect system concept, either a dual mode engine which operates as a heat exchanger turbofan or a JP-4 combustion driven turbofan, or strictly a dedicated nuclear engine may be used.

The direct engine system was considered in two different configurations. One configuration uses a gas turbine generator located in each engine and the other uses two gas turbine generators centrally located in the fuselage. For the central turbine configuration, power is transmitted to the engines via shafts and gear trains housed in the wings. Within this concept, the Brayton cycle and the Brayton cycle with regeneration were evaluated.

A necessary subsystem considered was a gas turbine generator-pump used to pump the helium through the reactor primary coolant loops. A similar subsystem was evaluated for pumping helium gas through the secondary coolant loops to the engines.

A block diagram depicting the organization of the overall engine systems study is shown in Figure 1.5-1.

An example of the effects of the engine parameters investigated on net thrust of the heat exchanger engine is summarized in Table 1.5-1. This table is for the base engine which has a bypass ratio of 4.0, engine core mass flow of 200 lbs/sec, and power extraction of 100 horsepower. Similar analysis was performed for several higher bypass ratios and is discussed in Section 6. From Table 1.5-1 it may be seen that a 5.1% change in turbine inlet temperature gives the greatest (8.2%) change in net thrust. Changing the bypass ratio will result in a different maximum net thrust at a different fan pressure ratio. A 16.7% decrease in overall engine pressure ratio will net a slight (1.9%) increase in thrust and a small (10%) reduction in compressor weight; however, to maintain the turbine inlet temperature, the reactor would have to be more powerful and therefore heavier. The increase in reactor weight offsets any engine weight saved.

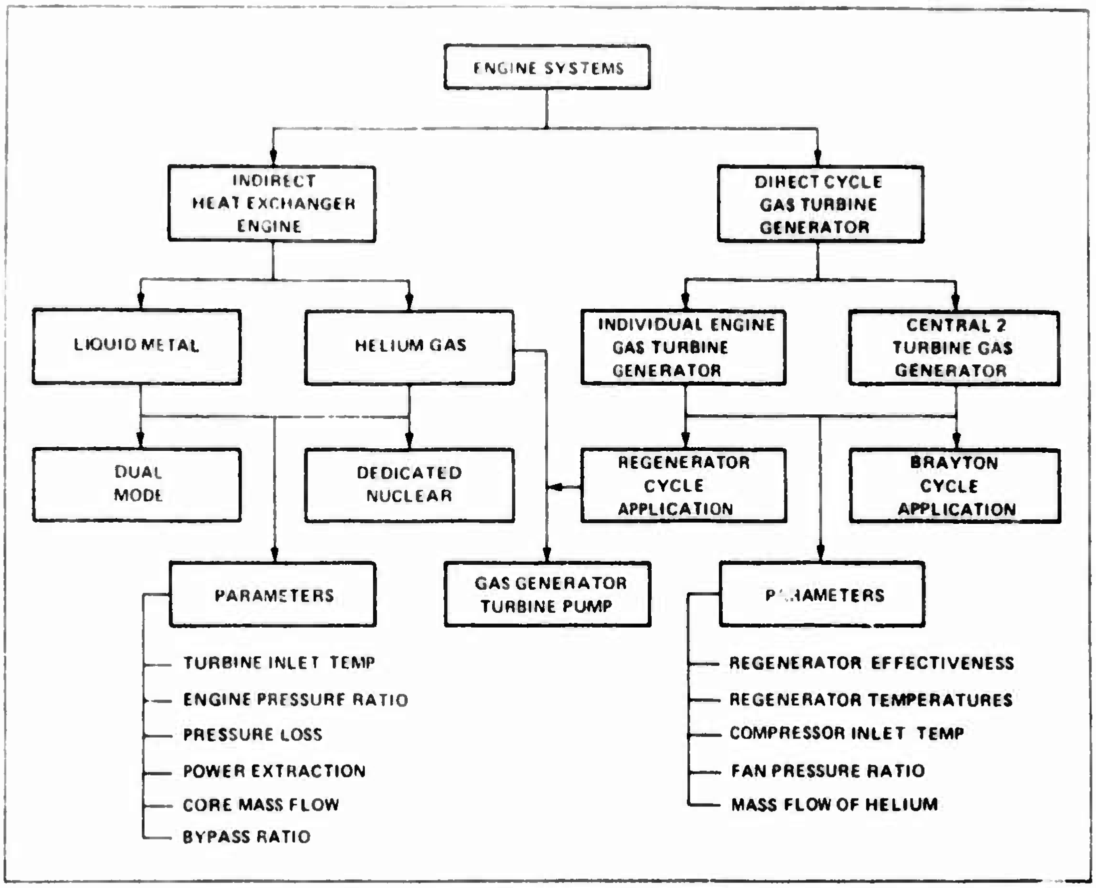


Figure 1.5-1. Engine Systems Study Diagram

TABLE 1.5-1. EFFECTS OF ENGINE PARAMETER VARIATIONS ON THE NET THRUST OF THE HEAT EXCHANGER TURBOFAN ENGINE

BASE ENGINE BYPASS RATIO 4.0 ENGINE CORE MASS FLOW 200 LBS/SEC POWER EXTRACTED 100 HP NET THRUST 14,580 LB			
PARAMETER	BASE	% CHANGE IN PARAMETER	% CHANGE IN NET THRUST
FAN PRESSURE RATIO	1.4	+ 14.3	+3.05
HORSEPOWER EXTRACTED	100.0	+900.0	–5.50
PRESSURE LOSS IN CORE%	11.0	+ 45.5	–2.36
TURBINE INLET TEMPERATURE, R	1960	+ 5.1	+8.20
OVERALL ENGINE PRESSURE RATIO	16.8	– 16.7	+1.90

The regenerative cycle produced 11 to 25% more work (BTU/lb) than the standard Brayton cycle for the direct system concept. A detailed discussion of these two cycles and their performance is presented in Volume I, Sections 6.2.4 and 6.2.5. Increasing the regenerative effectiveness from 0.75 to 0.90 would reduce the reactor output requirement by approximately 10%; however, the regenerator weight and volume increase with increasing effectiveness.

Table 1.5-2 gives a brief summary of several systems that resulted from this study.

TABLE 1.5-2. ENGINE SYSTEM STUDY SUMMARY

TYPE ENGINE HORSEPOWER EXTRACTED	HELIUM 1300	HELIUM 200	LIQUID-METAL 500
INDIRECT ENGINE SYSTEM			
BYPASS RATIO	4.0	6.0	5.5
NET THRUST (LB)	12,600	15,200	14,750
NUMBER OF ENGINES	12	10	10
THRUST/MW	254	320	310
ENGINE WEIGHT (LBS)	13,680	16,800	15,600
TURBINE-PUMP WT.	11,000	18,000	—
TUR. PUMP MW REQ.	61.2	95.2	—
TOTAL MW	636	574	475
TOTAL ENG. SYSTEM WT.	18,700	181,000	160,000
DIRECT ENGINE SYSTEM			
NET THRUST (LB)	14,000	11,950	
NUMBER OF ENGINES	10	12	
THRUST/MW	220	206	
ENGINE WEIGHT (LBS)	7,111	6,714	
TURBINE-PUMP WT.	12,000	10,000	
TUR. PUMP MW REQ.	63.6	58	
TOTAL MW REQ.	700	754	
TOTAL ENG. SYSTEM WT.	83,110	90,568	

1.6 HEAT TRANSFER (Volume I, Section 7): Three different heat transfer systems were analyzed. Two systems were designed for the indirect cycle engines, one using NaK as a working fluid, and the other using helium gas. A third system was designed for the helium direct cycle engines.

All three systems were designed with similar cross flow, plate-fin reactor heat exchangers, with an effectiveness of 0.75. The reactor heat exchangers for the gas systems used a plate spacing of 0.1 in. with 46 fins/in. and weighted a total of 51,000 lbs. The liquid metal reactor heat exchanger used a plate spacing of 0.256 in. with 16.96 fins/in. and weighed a total of 6500 lbs.

Both concentric and parallel pipe configurations were analyzed. In the liquid metal system the type of pipe made no significant difference in weight, but in the gas systems a savings of over 50,000 lbs was possible using the concentric pipes (65,000 lbs vs 12,000 lbs). Although the concentric pipe arrangement offers a weight savings, it presents more problems in the areas of fabrication and inner pipe stability under flow conditions, and additional pumping power. This configuration should be explored in greater depth to determine the magnitude of these problems and their solutions.

The maximum temperature allowed by the design parameters in the heat transfer system was 1800°F. The temperature used was the present state-of-the-art limit in metals designed to operate for long periods at high temperatures. This temperature was used rather than a higher temperature based on anticipated technological breakthroughs in metallurgy. Above 1800°F, the creep-rupture stress value is degraded to the point where the required pipe thicknesses become very large. Consequently, the pipe weight, even in the concentric configuration, becomes impractical. Further study and experimentation in high temperature metallurgy should be accomplished with the goal of increasing the allowable long duration working temperatures.

Two pass cross-counterflow finned-tube heat exchangers were used in the indirect cycle engines. The liquid metal heat exchanger, which was designed by the General Electric Company, Energy Systems Programs, is arranged in a wraparound configuration and has 4032 finned tubes and weighs 9300 lbs. The gas engine heat exchanger has two heat exchanger sections, each one a mirror image of the other, arranged in-line. This was necessary to get the 9792 tubes required within the diameter of the engine. The heat exchanger weighs 20,600 lbs. Section 7.4.1 (Volume I) gives a more complete discussion of these heat exchangers.

The direct cycle engines require two heat exchangers: a recuperator and a precooler. Each recuperator is similar in design to the reactor heat exchanger and weighs 3050 lbs. The precoolers were designed using the same procedures as the indirect cycle engine heat exchangers. They are arranged in a rectangular configuration as opposed to the wraparound arrangement of the engine heat exchangers. Each precooler weighs 10,500 lbs.

The pumping power required to circulate the two different working fluids is quite different. Liquid metal may easily be pumped at a small cost in horsepower while helium gas required a complex turbine circulation system. In addition, the two different helium systems have different pumping requirements as the direct cycle system is self pumping, while the indirect cycle is not. Both gas systems require a primary loop circulator system of approximately 25,000 hp. The circulator is similar to a direct cycle engine with a helium gas powered turbine turning the circulator pump. This adds over 11,000 lbs to either gas system in additional heat exchanger weight.

The indirect cycle helium system requires a similar, though less powerful, circulator for its secondary or engine loop. (The reactor core has a much higher pump power requirement than does either heat exchanger.) The extra heat exchangers required add 8100 lbs to the system weight.

These gas circulation systems not only add heat exchanger and pump weight, they also require additional power from the reactor which necessitates a larger, heavier reactor. Table 1.6-1 shows a comparison of the system power required with and without the gas circulators. Refer to Volume I, Section 5 of this report for the weight comparison.

TABLE 1.6-1. GAS CIRCULATION POWER REQUIREMENTS

	POWER REQUIRED FOR ENGINES (MW)	POWER REQUIRED FOR ENGINES & CIRCULATORS (MW)
INDIRECT CYCLE HELIUM GAS SYSTEM	475	574
DIRECT CYCLE HELIUM GAS SYSTEM	636	700

Safety isolation valves are required on all working fluid lines entering and leaving the reactor containment vessel. These valves can add considerable weight to the heat transfer system since there can be as many as 22 pipes going from and to the reactor (indirect cycle, 2-pipe configuration for 10 engines with one circulator pump)

Three valve designs are presented, one by Sandia Corp. and two designs proposed by this study. All the valves are designed to be constructed from Haynes Alloy 188, close in 50 msec, and withstand an axial decelleration of 1500 g's without opening. A summary of weights including safety valves, and reactor power requirements of the three heat transfer systems is presented in Table 1.6-2.

TABLE 1.6-2. WEIGHT AND POWER COMPARISONS

	INDIRECT CYCLE		
	LIQUID METAL	HELIUM	DIRECT CYCLE
WEIGHT (LBS)			
HEAT EXCHANGERS	99,500	257,600	176,300
VALVES	136,000	81,600	74,800
PIPES	12,100	12,200	12,200
COOLANT	27,000	100	100
TOTAL	274,600	351,500	263,400
REACTOR POWER (MW)	475	574	700

1.7 SAFETY (Volume I, Section 8): Three categories of accidents were used to compute the probability of releasing radioactive material to the environment. These were the Crash Acci-

dent, which considers a mid-air collision or ground impact, the Transient Accident which considers all other accidents that result in a demand for reactor shutdown, except for total loss of coolant, and total Loss of Coolant Accident.

The total Loss of Coolant Accident was found to have a probability of occurrence of less than 10^{-25} per flight and was excluded from further consideration because the probability of occurrence was insignificant in comparison to the other accident categories.

The results of the Crash and Transient Accident analyses are summarized in Table 1.7-1. As shown, there is 90% confidence that the probability, per flight, of a release of radioactive material lies between 1.87×10^{-3} and 4.43×10^{-3} with a mean, or expected value, of 3.18×10^{-3}.

TABLE 1.7-1. TOTAL RELEASE PROBABILITY PER FLIGHT

ACCIDENT CATEGORY	PROBABILITY			
	0.05	0.50	0.95	MEAN
CRASH	1.86×10^{-3}	3.18×10^{-3}	4.43×10^{-3}	3.18×10^{-3}
TRANSIENT	1.97×10^{-9}	4.90×10^{-8}	1.03×10^{-6}	2.96×10^{-7}
TOTAL	1.87×10^{-3}	3.19×10^{-3}	4.43×10^{-3}	3.18×10^{-3}

The crash accident dominates the overall probability of incurring a release of radioactive material, and for that reason, the consequence of a release, in terms of fatalities, was modeled for the crash accident only, using a 574 MW reactor.

The results of that analysis, summarized in Table 1.7-2, show the number of fatalities resulting from a release of radioactive material. There is a 99% confidence that the number of fatalities lies between 0.003 and 865. The mean, or expected number, is 18.9 fatalities per release of radioactive material.

TABLE 1.7-2. FATALITIES PER RELEASE OF RADIOACTIVE MATERIAL PROBABILITY DISTRIBUTION

	DEATHS PER RELEASE			
PROBABILITY	0.005	0.50	0.995	MEAN
DEATHS	0.003	1.20	865	18.9

Table 1.7-3 shows a similar analysis, but is given in fatalities per year based on the number of flights per year. For example, for 1000 flights per year, the expected number of fatalities is 17.8.

The individual risk to any member of society was expressed as the expected number of deaths divided by the total U.S. population. For an expected number of fatalities of 17.8 per year, and a population of 201 million, the individual risk is 8.9×10^{-8} per year. Stated another

TABLE 1.7-3. DEATHS PER YEAR PROBABILITY DISTRIBUTION

	DEATHS PER YEAR			
FLIGHTS PER YEAR	**0.005**	**0.50**	**0.95**	**MEAN**
1000	**<1**	**1.13**	**814**	**17.8**
1500	**<1**	**1.18**	**850**	**18.5**
2000	**<1**	**1.19**	**860**	**18.8**

way, each individual has approximately 1 chance in 11 million, each year, of being killed by a release of radioactive material from a nuclear powered airplane over the continental United States.

1.8 LIFE CYCLE COST (Volume I, Section 9): This study undertook the task of determining the life cycle cost for a nuclear powered aircraft. The life cycle cost for the aircraft included flyaway cost, initial spares, peculiar support and training equipment, research, development, test and evaluation, and, finally, operating and support cost discounted over 10 years. Of the proposed systems, only the 2,000,000 lb aircraft with a dedicated propulsion system was addressed. The study determined the life cycle cost of 60 production airframes and one prototype airframe through the use of cost-estimating relationships, the cost of analogous items, and a best estimate of the cost presented by experts associated with the items in question. The flyaway cost was determined to be in the range of $127 - 224 million. Figure 1.8-1

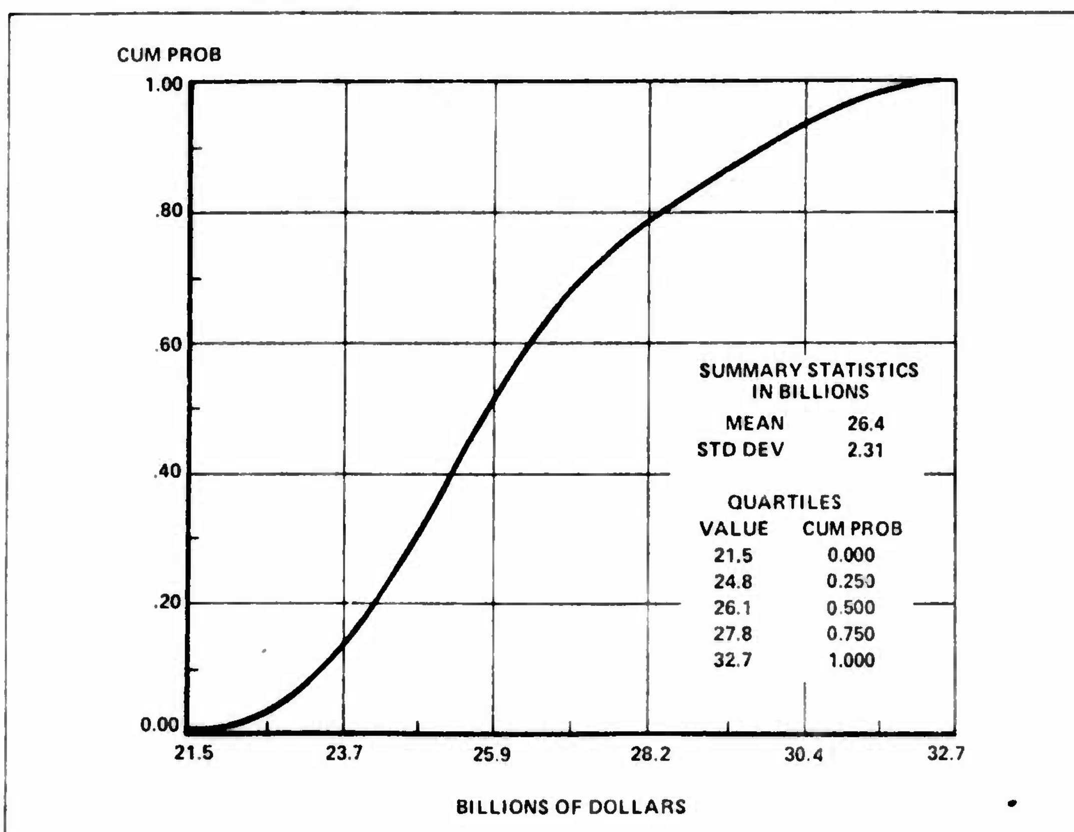


Figure 1.8-1. Life Cycle Cost for 60 Nuclear Aircraft - Cumulative Probability

illustrates the range of cost for the life cycle of 60 nuclear powered aircraft, i.e., from the birth to the death of the system. Its interpretation is that if a nuclear aircraft program were undertaken the life cycle cost would lie in the range of $21.5 - 32.7 billion. The mean or expected life cycle cost would be $26.4 billion with the median being $26.1 billion in 1974 dollars.

2 WEAPON SYSTEM

2.1 PROPULSION SYSTEM: In the past several years, there has been no large scale integrated systems study performed on an airborne nuclear propulsion system. There have, however, been several studies performed concerning airborne applications of several components of a nuclear propulsion system such as the reactor and dual mode engines. Each of these studies attempted to optimize about one or more parameters peculiar to the component being analyzed. It is important to realize that misleading results can be obtained by this type of subsystem analysis, and that individual components may be less than optimum when the overall nuclear propulsion system is optimized.

The nuclear propulsion system designs in this report utilized reactor data from the Westinghouse Electric Corporation, Astronuclear Division, and a doctoral dissertation prepared by K. L. Mills and then used a systems approach to design the engines, heat exchangers, piping, valves, and pumping systems required to complete the propulsion system.

Shown in Table 2.1-1 are the propulsion system weights resulting from the eight possible combinations of the reactors, working fluids, and engine types. These weights represent the total propulsion system weights necessary to provide the required cruise thrust for the point design aircraft.

It can be seen that the engines and heat transfer systems are the lightest in the helium-cooled, direct cycle systems, but the lightest gas reactor (442,000 lbs) is the Mills reactor which weighs 78,000 lbs less than the Westinghouse liquid-metal reactor. However, three points must be considered when comparing the Mills and Westinghouse reactor designs:

1) the Westinghouse reactor has a 10,000 hr lifetime compared to the Mills' 3000 hr lifetime. Note that this difference in reactor lifetime will affect other parameters associated with the propulsion system. For instance, the cost data computed in the analysis are based on the 10,000 hr lifetime reactor and do not reflect any changes due to the increased number of reactor refurbishings required by a 3000 hr reactor.

2) The Mills reactor lateral radiation level is about five times that of the Westinghouse reactor.

3) The Mills reactor is a one man design effort and thus, probably, does not incorporate the depth of engineering contained in the Westinghouse effort.

Before a final comparison can be made, however, the total system must be examined with the aircraft and nuclear propulsion system integrated.

TABLE 2.1-1. NUCLEAR PROPULSION SYSTEMS

	PROPULSION SYSTEM							
	1	2	3	4	5	6	7	8
THRUST/ENGINE (1000 LB)	14.7	14.3	15.2	14.7	15.2	14.7	14.0	14.0
NUMBER OF ENGINES	10.0	10.0	10.0	10.0	10.0	10.0	10.0	10.0
TOTAL ENGINE WEIGHT (1000 LBS)	156.0	164.0	181.0	188.0	181.0	188.0	83.0	83.0
HEAT TRANSFER SYSTEM WEIGHT (1000 LBS)	275.0	275.0	351.0	351.0	351.0	351.0	270.0	270.0
REACTOR POWER (MW)	475.0	475.0	574.0	574.0	574.0	574.0	700.0	700.0
REACTOR WEIGHT (1000 LBS)	520.0	520.0	719.0	719.0	442.0	442.0	789.0	494.0
TOTAL PROPULSION SYSTEM WEIGHT (1000 LBS)	951.0	959.0	1251.0	1258.0	974.0	981.0	1142.0	847.0

PROPULSION SYSTEM DESCRIPTIONS

1 - NaK / INDIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
2 - NaK / INDIRECT CYCLE / DUAL MODE ENGINES / WESTINGHOUSE REACTOR
3 - HELIUM / INDIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
4 - HELIUM / INDIRECT CYCLE / DUAL MODE ENGINES / WESTINGHOUSE REACTOR
5 - HELIUM / INDIRECT CYCLE / DEDICATED ENGINES / MILLS REACTOR
6 - HELIUM / INDIRECT CYCLE / DUAL MODE ENGINES / MILLS REACTOR
7 - HELIUM / DIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
8 - HELIUM / DIRECT CYCLE / DEDICATED ENGINES / MILLS REACTOR

2.2 TOTAL AIRCRAFT SYSTEM: The point design, 2,000,000 lbs gross weight, canard configuration aircraft developed in Volume I, Section 4 and summarized in Section 1.3 was coupled with the six possible propulsion system configurations given in Section 2.1 to yield a variety of possible payloads. Specifically, the alternatives include considerations of conventional aircraft construction versus a lighter weight advanced composite construction; pure chemical power for takeoff versus chemically augmented nuclear power; and the eight proposed nuclear propulsion systems.* These alternatives yield 32 possible combinations, the results of which, in terms of available payload, are summarized in Table 2.2-1.

Table 2.2-1 shows the following major points.

1) A positive payload for a 2,000,000 lbs aircraft cannot be achieved under any of the propulsion system options using conventional aircraft construction and a purely chemical takeoff.

*The chemically augmented, nuclear powered takeoff uses nuclear powered engines plus dedicated chemical engines. The nuclear engines produce cruise thrust corrected from the design point altitude conditions to the ambient air conditions at the departure field. The dedicated chemical engines produce the remainder of the thrust required for takeoff.

2) A positive payload can be achieved under all the propulsion systems options using advanced composite construction and chemically augmented nuclear takeoff.

3) The highest payload is obtained using the Mills reactor in a direct cycle with advanced composite construction and chemically augmented nuclear takeoff.

4) From a payload standpoint, the worst combination exists when using the Westinghouse reactor in a helium indirect cycle with conventional aircraft construction and a pure chemical takeoff capability.

5) In all propulsion system options, the NaK cooled indirect cycle produces a larger payload than the helium cooled indirect cycle.

6) The direct cycle helium system, when compared with the indirect cycle, produces a larger payload in all aircraft configurations.

7) Using a helium cooled Westinghouse reactor in either direct or indirect cycle, it is necessary to go to advanced composite construction and chemically augmented nuclear takeoff to produce a positive payload.

8) Under all propulsion system options, using chemically augmented nuclear power for takeoff yields a higher payload than pure chemical power for takeoff.

Whereas Table 2.2-1 compares the different propulsion systems from a payload standpoint, safety considerations must also be taken into account in the overall evaluation. Following is a list of some major points that must be considered:

1) The helium system operates under high pressure (1800 psi) in both the primary and secondary heat transfer loops whereas the NaK system operates at a low pressure (130 psi).

2) The corrosion problems encountered in the NaK system are much more severe than in the helium system.

3) The atomic structure of NaK allows it to become activated, whereas the helium is radioactively inert.

4) While high pressure gas and explosively reacting NaK may be deemed comparably hazardous from a crew safety standpoint, NaK would be much more damaging to the general populace and the environment in the event of a crash.

TABLE 2.2-1. PREDICTED PAYLOAD FOR 2,000,000 LBS GROSS WEIGHT AIRCRAFT

PAYLOAD $\times 10^3$LB / PAYLOAD/GROSS WT.

PROPULSION SYSTEM	AIRCRAFT CONFIGURATION A: CONVENTIONAL CONSTRUCTION/ PURE CHEMICAL TAKEOFF	B: CONVENTIONAL CONSTRUCTION CHEMICALLY AUGMENTED NUCLEAR TAKEOFF	C: COMPOSITE CONSTRUCTION/ PURE CHEMICAL TAKEOFF	D: COMPOSITE CONSTRUCTION/ CHEMICALLY AUGMENTED NUCLEAR TAKEOFF	
1	-190 / 0	90 / 0.05	120 / 0.06	400 / 0.20	NaK / INDIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
2*	-146 / 0	90 / 0.05	160 / 0.08	400 / 0.20	NaK / INDIRECT CYCLE / DUAL MODE ENGINES / WESTINGHOUSE REACTOR
3	-484 / 0	-214 / 0	-174 / 0	96 / 0.05	HELIUM / INDIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
4*	-444 / 0	-204 / 0	-134 / 0	106 / 0.05	HELIUM / INDIRECT CYCLE / DUAL MODE ENGINES / WESTINGHOUSE REACTOR
5	-212 / 0	68 / 0.04	98 / 0.05	378 / 0.19	HELIUM / INDIRECT CYCLE / DEDICATED ENGINES / MILLS REACTOR
6*	-114 / 0	126 / 0.06	196 / 0.10	436 / 0.22	HELIUM / INDIRECT CYCLE / DUAL MODE ENGINES / MILLS REACTOR
7	-380 / 0	-100 / 0	-70 / 0	210 / 0.11	HELIUM / DIRECT CYCLE / DEDICATED ENGINES / WESTINGHOUSE REACTOR
8	-84 / 0	196 / 0.10	176 / 0.09	506 / 0.25	HELIUM / DIRECT CYCLE / DEDICATED ENGINES / MILLS REACTOR

*CHEMICAL ENGINE WEIGHT REMOVED FOR DUAL MODE ENGINE SYSTEMS

2.3 **MISSION APPLICATION:** In a chemically powered turbofan engine, the heat is manufactured in the engine by the burning of fuel, whereas in a nuclear engine the heat source is a remotely located reactor. The necessary heat transfer system with at least two heat exchangers is complicated and inflexible with regard to air temperature changes. Changes in altitude, and therefore air temperature, require the indirect cycle engine to vary the nozzle and fan pitch in order to maintain the constant airflow parameters required as the heat exchanger input conditions. If these airflow input conditions are not maintained constant, the heat exchanger is no longer adequate, and the turbine inlet temperature will drop resulting in a loss of thrust. The direct cycle engine has virtually identical problems; it may be designed for a lower altitude and the air flow to its heat exchanger varied, but failing to do so leads to an engine which does not produce the design point thrust.

The mechanical complexity needed to provide constant airflow parameters can alternately be avoided by using chemical power for the off-design altitude portion of the mission. However, this again restricts the aircraft because of the following:

1) The aircraft must replace the chemical fuel consumed in altitude changes. This would require that either the aircraft curtail its mission length and return to base, or that it be refueled in flight.

2) Any supplementary chemical fuel load carried subtracts directly from the payload.

For these reasons, a mission which requires changes in altitude may be less suitable for the application of nuclear power than one which requires that the entire mission be flown at a constant altitude. Table 2.3-1 shows the candidate mission requirements for the nuclear aircraft. The above altitude restrictions may severely restrict the nuclear powered aircraft's

TABLE 2.3-1. AIRCRAFT PARAMETERS FOR SELECTED MISSIONS

PARAMETER	MISSION				
	ASW	AWAC	C/C	Mx	CARGO
MISSION DURATION	DAYS	DAYS	DAYS	DAYS	HOURS
CREW SIZE (NO. OF MEN)	FLT 4-5 MISSION 6-8	FLT 4-5 MISSION 17	FLT 4-5 MISSION 19	FLT 4-5 MISSION 4-6	FLT 4-5 MISSION 1
AIRSPEED	SUBSONIC	SUBSONIC	SUBSONIC	SUBSONIC	HIGH SUBSONIC
ALTITUDE (FT)	0 to 25,000-35,000	25,000-35,000	25,000-35,000	25,000-35,000	25,000-35,000
PAYLOAD (LBS)	130,000-200,000	91,000	200,000+	400,000	200,000-400,000
FLEET SIZE (NO. OF AIRCRAFT)	60	34	17	25-50	40-81
RANGE/ ENDURANCE	UNLIMITED	UNLIMITED	UNLIMITED	UNLIMITED	UNLIMITED

A COMPLETE EXPLANATION OF THE PARAMETERS IS CONTAINED IN VOLUME III.

ASW mission effectiveness. A choice of a mission flown at constant altitude would allow the propulsion system designer to optimize the system at that altitude. Thus, airborne warning and control (AWAC) or command and control (C&C) may be more viable candidates. However, both of these missions require extended flight times over populated areas and thus imply increased concerns with respect to safety of the general populace. The AWAC and C&C may require only a few hundred miles of emergency chemical fuel capability. These missions can be flown near their home station and will not require the extended over water range nor require the large chemical fuel reserve that the ASW mission does. This fact results in potentially larger payloads for the AWAC and C&C missions using the same point design aircraft.

3 CONCLUSIONS AND RECOMMENDATIONS

The conclusions and recommendations contained in this section of the report are related, primarily, to considerations of system weights and safety, and, as such, represent only the major conclusions and recommendations derived from the nuclear airplane feasibility study. Numerous other conclusions and recommendations may be found in the various sections of the main text.

Although the individual sections of the main report are accurate and largely complete in their own right, it is, nonetheless, true that false or misleading conclusions regarding the feasibility of the nuclear airplane concept can be drawn by considering only isolated portions of this study. It is, therefore, recommended that users of this report consider the report in its entirety before making judgments or decisions regarding nuclear propulsion for aircraft.

3.1 AVIONICS: The avionics portion of this study has shown the requirements that are necessary to achieve a specified mission reliability through the use of equipment redundancy only. For example, to achieve 90% reliability requires approximately 4 levels of redundancy for a 96 hr mission, and approximately 10 levels of redundancy for a 336 hr mission.

It was concluded that, while not absolutely essential, alternatives to redundancy for achieving high reliability are desirable, and it is recommended that the option of utilizing an in-flight repair capability be studied to determine its feasibility.

3.2 REACTOR: The reactor analysis utilized two reactor design proposals by the Westinghouse corporation and one design proposal by K. L. Mills in assessing the nuclear airplane concept. Because of the 200 ft minimum separation between the crew and the reactor, which results from the canard configured aircraft used in this study, it was concluded that a potential weight savings exists, through reduced radiation shielding, for all three reactor designs. Reactor weight savings of 10,000 to 21,000 lbs are possible without exceeding the original design specifications for maximum allowable radiation dose rate to the crew. It is, therefore, recommended that any future designs consider the use of large separation distance between the crew and the reactor, and that the reactor be designed for that specific application.

It was also concluded that reactor weight savings of up to 300,000 lbs can be realized through the use of increased power density fuels. Therefore, it is recommended that the

development of higher power density fuels be considered as part of any future nuclear aircraft development programs.

Following a reactor shutdown, heat continues to be generated by the decay of radioactive fission products. This reactor afterheat could be as high as 25 million BTU per hr, 30 min after shutdown of a 574 MW reactor. Although this problem was not specifically addressed in this study, it was concluded that a means of removing reactor afterheat during ground operations must be devised.

3.3 NUCLEAR PROPULSION SYSTEM: The nuclear propulsion system analysis considered three propulsion system concepts. Both liquid metal and gas-cooled systems were considered, and, for the gas system, both direct and indirect cycle engines were considered. Of these three concepts, the liquid metal-cooled system is considerably lighter. This results from two facts: (1) the reactor, which is the heaviest part of the system, is inherently lighter for a liquid metal coolant than for a gas coolant, and (2) the engines and heat transfer systems for a gas cooled system are inherently less efficient than for a liquid metal system, thus requiring an even larger reactor.

Shown in Table 3.3-1 are some of the weight and power requirements for the three propulsion concepts. It can be seen that, if only the engine weights are considered, the direct cycle helium system is lightest at 83,000 lbs. A similar conclusion is drawn when considering only the heat transfer system where the direct cycle helium system weighs 242,000 lbs. However, when considering only the reactor, the liquid metal system, at 520,000 lbs is substantially lighter than either of the helium systems.

TABLE 3.3-1. PROPULSION SYSTEM WEIGHTS AND POWER REQUIREMENTS (2,000,000 LBS AIRCRAFT)

TYPE OF SYSTEM	WEIGHT (LBS)				POWER (MW)	
	ENGINE[1]	HEAT TRANSFER[2]	REACTOR[3]	TOTAL	PUMP	TOTAL
HELIUM DIRECT CYCLE	83,000	242,000	810,000	1,035,000	64	700
INDIRECT CYCLE	181,000	323,000	740,000	1,244,000	95	574
LIQUID METAL	156,000	267,000	520,000	943,000	1.5	475

The total propulsion system weights of 1,035,000 lbs for the direct cycle helium system, 1,244,000 lbs for the indirect cycle helium system, and 943,000 lbs for the liquid metal system show the liquid metal system to be substantially lighter than either helium system.

Additionally, if the efficiencies of the helium systems were assumed to equal that of the liquid metal system, so that all systems could use a 475 MW reactor, the reactor for the helium cooled systems would weigh 730,000 lbs versus the 520,000 lbs reactor for liquid metal. In such a case, a helium-cooled system would still weigh at least 112,000 lbs more than a liquid metal-cooled system. Thus, it is concluded that a liquid metal-cooled propulsion system will be substantially lighter than a helium-cooled propulsion system.

However, the relative safety of liquid metal vs gas systems is a question that has not been addressed in this study, and it is recommended that such a study be accomplished before any commitment is made to develop a nuclear powered airplane.

3.4 CHEMICAL PROPULSION SYSTEM: The efficiency of a nuclear propulsion system is such that it will not develop a sufficient thrust for takeoff. Therefore, it is necessary that a chemical fuel propulsion system be incorporated in the aircraft design.

In this design study, two chemical propulsion system requirements were defined for point design calculations. In the first case, it was assumed that only chemical power would be used for takeoff and landing, with an additional requirement for a 1000 nm emergency cruise capability. These criteria led to a chemical propulsion system weighing 320,000 lbs, or 16% of the gross weight of the 2,000,000 lbs point design aircraft.

In the second case, the chemical propulsion system was only required to augment the nuclear propulsion system for takeoff and landing, with no requirement for emergency cruise capability. In this case, the chemical propulsion system weighed 80,000 lbs, or 4% of the aircraft gross weight.

The 80,000 lbs system represents the minimum weight that must be allocated to chemical propulsion requirements since it only supplies the minimum thrust augmentation necessary for takeoff and landing. Additional requirements for chemical fuel capability must be evaluated in terms of the additional weight imposed.

3.5 PAYLOAD: A 2,000,000 lbs aircraft, constructed by conventional construction techniques, and carrying sufficient chemical fuel for takeoff, landing, and emergency cruise (320,000 lbs) cannot achieve a positive payload with any of the nuclear propulsion systems considered in this study. To achieve the required ASW payload (200,000 lbs) under these criteria would require an aircraft with a gross weight in excess of 3,000,000 lbs, assuming that the estimation techniques used in this study are valid for gross weights exceeding 2,000,000 lbs.

A 2,000,000 lbs aircraft, constructed by conventional construction techniques, but with the chemical fuel capability reduced to the minimum necessary for takeoff and landing (80,000 lbs) cannot achieve a positive payload using a helium-cooled nuclear propulsion system. A payload of 90,000 lbs could be achieved by using a liquid metal system.

Expert predictions indicate that airframe weight savings as high as 40% may be realized by advanced composite construction techniques. Based on these predictions, a 2,000,000 lbs aircraft, with a 320,000 lbs chemical fuel system, still cannot achieve a positive payload using a helium-cooled nuclear propulsion system. A payload of 120,000 lbs could be achieved by using a liquid metal system.

A 2,000,000 lbs aircraft, constructed by advanced composite construction techniques, and utilizing an 80,000 lbs chemical fuel system can achieve a positive payload of 96,000 to 210,000 lbs with a helium-cooled nuclear propulsion system, and 400,000 lbs with a liquid metal system.

It is concluded that, in any case, a liquid metal-cooled nuclear propulsion system will yield the largest payload. Additionally, the predicted weight savings to be offered by advanced composite construction techniques indicates that such technology should be pursued if the nuclear airplane is to be feasible. Also, it should be noted that the payload predictions given in this section have incorporated the reactor weight savings discussed previously in Section 3.2.

It is recommended that, rather than going to increasingly larger aircraft, a concerted effort be made to reduce both propulsion system and airframe weights in order to make the nuclear powered airplane a more viable concept.

3.6 SAFETY: This study has concluded that the overall probability of a release of radioactive material is dominated by the contribution from the crash accident. The probability of a release of radioactive material resulting from a crash is approximately three orders of magnitude higher than all other accidents, and is dominated by three factors. These are, in ascending order of probability, the probability of failure to seal the containment vessel at impact, the probability of failure of the containment vessel to withstand the impact, and the probability of failure of the containment vessel to withstand the afterheat transient.

To date, only one known study of safety valves for sealing containment vessels has been performed. It is recommended that further development of safety valves be accomplished with design objectives of lighter weight and higher reliability.

Previous tests of containment vessel impact survivability used smooth surfaced spheres, free of any structures. It is recommended that further evaluation of containment vessel impact survivability be accomplished with emphasis on the effects of the aircraft structure, valve weldments, pipe penetrations, and other protuberances.

Previous analyses of afterheat transient survival have assumed ideal insulation with no voids, pumps, pipes or heat exchangers. It is recommended that further analysis be performed with consideration given to the effects of the voids and the installed heat transfer equipment.

Of special importance to the survivability of the containment vessel during the afterheat transient is the depth of burial resulting from the crash impact. Previous studies have assumed burial depths for an unenclosed sphere impacted against soil. It is recommended that the effect of the aircraft structure on depth of burial be studied.

A release of radioactive material to the environment is not expected to result in a large number of fatalities. Given that a release occurred from a 574 MW reactor, there is a 95% confidence that the number of fatalities would be equal to or less than 52, with an expected number of 19. When a release of radioactive material does occur, it will necessitate the evacuation of a large area until the dose rate from radioactive particles returns to an acceptable level.

Historical aircraft crash data indicates that more than one-half of the crashes will occur in the vicinity of the airport. Therefore, it is recommended that further study be conducted into the safety and operational implications of having to close or evacuate an airport, for up to 90 days, after each crash.

Initial investigations, based strictly on historical data from ground based reactors, indicated a median probability of a reactor shutdown, due to some transient condition, of 0.3 per flight. It was concluded that such a shutdown rate would have an unacceptable impact on safety and mission reliability. Additionally, a premature reactor shutdown on every third flight would substantially increase the estimated life cycle costs from those summarized in Section 1.8, and the option of reducing the chemical fuel to the minimum needed for takeoff and landing could not be considered.

It is recommended that a detailed study of reactor systems be performed to insure an acceptable reliability.